Regolith Derived Heat Shield for Planetary Body Entry and Descent System with In Situ Fabrication

NASA Innovative Advanced Concept (NIAC)

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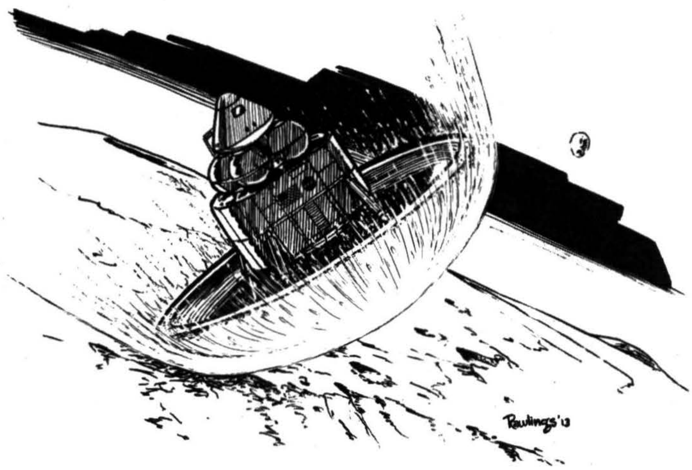

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Acronyms, Abbreviations, and Units

ARC	Ames Research Center, Moffett Field, CA
CIF	Center Innovation Fund
DOE	Department of Energy
DRA	Design Reference Architecture
EDL	Entry Descent & Landing
ESPL	Electrostatics & Surface Physics Laboratory, Kennedy Space Center
FSP	Fission Surface Power
GMRO	Granular Mechanics & Regolith Operations, Kennedy Space Center
GRC	Glenn Research Center
IHF	Interaction Heating Facility, ARC
IMLEO	Inserted Mass In Low Earth Orbit
IPA	Isopropyl Alcohol
IR	Infra-red
ISRU	In Situ Resource Utilization
KSC	Kennedy Space Center, FL
kW_e	Electrical Power Kilowatts
LEO	Low Earth Orbit
LLO	Low Lunar Orbit
LMO	Low Mars Orbit
MER	Mars Exploration Rover
MMRTG	Multi Mission Radioisotope Thermoelectric Generator
NE	Engineering & Technology Development Directorate, NASA, KSC
PICA	Phenolic Impregnated Carbon Ablator
PMAD	Power Management and Distribution
RTG	Radioisotope Thermoelectric Generator
RTV	Room Temperature Vulcanizing
SEP	Solar Electric Propulsion
t	Metric Tonne (1000 kg)
TC	Thermocouple
TPS	Thermal Protection System
Δv	Delta-v (change in velocity)

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I. Introduction

High-mass planetary surface access is one of NASA's Grand Challenges involving entry, descent and landing (EDL). During the entry and descent phase, frictional interaction with the planetary atmosphere causes a heat build-up to occur on the spacecraft, which will rapidly destroy it if a heat shield is not used. However, the heat shield incurs a mass penalty because it must be launched from Earth with the spacecraft, thus consuming a lot of precious propellant.

This NIAC project investigated an innovative approach to provide heat shield protection to spacecraft after launch and prior to each EDL thus potentially realizing significant launch mass savings. Heat shields fabricated *in situ* can provide a thermal-protection system for spacecraft that routinely enter a planetary atmosphere. By fabricating the heat shield with space resources from materials available on moons and asteroids, it is possible to avoid launching the heat-shield mass from Earth. Regolith has extremely good insulating properties and the silicates it contains can be used in the fabrication and molding of thermal-protection materials. Such in situ developed heat shields have been suggested before by Lewis [1]. Prior research efforts [2][3][4] have shown that regolith properties can be compatible with very-high temperature resistance. Our project team is highly experienced in regolith processing and thermal protection systems (TPS).

Routine access to space and return from any planetary surface requires dealing with heat loads experienced by the spacecraft during reentry. Our team addresses some of the key issues with the EDL of human-scale missions through a highly innovative investigation of heat shields that can be fabricated in space by using local resources on asteroids and moons. Most space missions are one-way trips, dedicated to placing an asset in space for economical or scientific gain. However, for human missions, a very-reliable heat-shield system is necessary to protect the crew from the intense heat experienced at very high entry velocities of approximately 11 km/s ~ Mach 33 (Apollo). For a human mission to Mars, the return problem is even more difficult, with predicted velocities of up to 14 km/s, ~ Mach 42 at the Earth-atmosphere entry. In addition to human return, it is very likely that future space-travel architecture will include returning cargo to the Earth, either for scientific purposes or for commercial reasons. Platinum, titanium, helium 3, and other metals, elements and minerals are all high-value commodities in limited supply on Earth, and it may be profitable to mine these substances throughout the Solar System and return them to Earth, if an economical method can be found. To date, several private corporations have been launched to pursue these goals. Because the heat shield is the last element to be used in an Earth-return mission, a high penalty is paid in the propellant mass required to carry the heat shield to the destination and back. If the heat shield could be manufactured in space, and then outfitted on the spacecraft prior to the reentry at Earth, then significant propellant and mass savings could be achieved during launch and space operations.

Preliminary mission architecture scenarios are described, which explain the potential benefits that may be derived from using an in-situ fabricated regolith heat shield. In order to prove that this is a feasible technology concept, this project successfully fabricated heat shield materials from mineral simulant materials of lunar and Martian regolith by two methods: 1) Sintering and 2) Binding the simulant with a "room-temperature vulcanizing" (RTV) silicone formulated to withstand high temperatures. Initially a third type of fabrication was planned using the hot waste stream from regolith ISRU processes. This fabrication method was discarded since the resulting

samples would be too dense and brittle for heat shields. High temperature flame tests at KSC and subsequent arc jet tests at Ames Research Center (ARC) have proved promising. These coupon tests show favorable materials properties and have the potential to be a new way of fabricating heat shields for space entry into planetary atmospheres

II. Proposed Mission Architectures

If an all-propulsive architecture is used to land a human mission on Mars then the gear ratio for Low Earth Orbit (LEO) to the Mars surface is approximately 20:1 [5]. In other words, one tonne (t) of landed mass on Mars would require 20 t in LEO. NASA Mars mission studies have proposed landers ranging from 40–60 t. For a 60 t lander this would amount to 2,000–3,000 t in LEO, which would require 20–30 heavy lift 100 t launchers to accomplish, and the on-orbit assembly would provide substantial risk and technical challenges. By contrast, if the Mars atmosphere is used to aerobrake the spacecraft and reduce its velocity, then only 5–6 t [5] are required in LEO for 1 t on the Mars surface. This would translate to 200–360 t in LEO, a much better proposition, but still too high to launch with one heavy lift 100 t launcher. Therefore we propose the use of in-situ fabricated, regolith-derived heat shields that can be made locally (in space) and then outfitted to the spacecraft prior to Mars EDL, to avoid launching this mass from Earth to LEO.

The use of silicate-rich regolith in the manufacturing of heat shields in space is only applicable to the entry into oxidizing atmospheres such as that of Earth and Mars. The hydrogen-rich atmospheres typical of the giant planets of the Solar System would quickly reduce silicates and all metal oxides in the regolith to its metallic elements and mostly vaporize the entire heat shield. Three mission Design Reference Architectures (DRA) are being studied for using this technology. One DRA is based on building the regolith heat shield using materials from the moons Phobos and Deimos for either descent onto Mars or return to Earth. Another is to build the heat shields on or in the vicinity of the Moon for an Earth return or a Mars mission. A third architecture envisions building heat shields on or near an asteroid for Earth return.

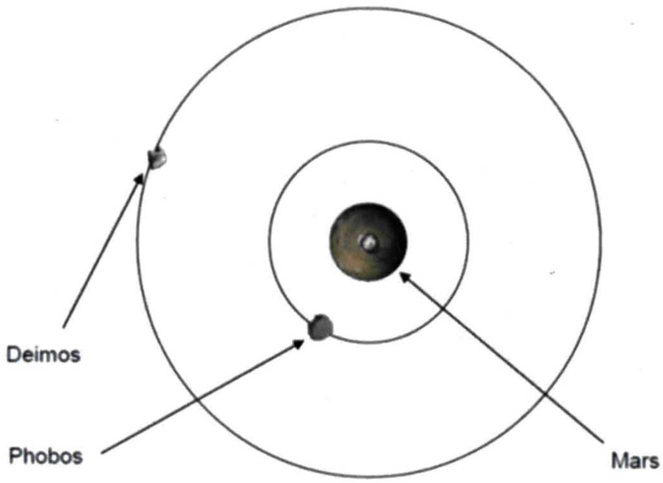
Architecture software tools such as ModelCenter™ by Phoenix Integration, Inc. are being evaluated to do the detailed mission calculations necessary. Cost and Δv calculations are being performed using tools such as Mathcad™ and Microsoft EXCEL™.

A. *Architecture I – In Situ Heat Shield Fabrication at Phobos/Deimos for Earth-bound Mars Return Spacecraft and Mars EDL of Surface Exploration Craft*

The NASA Mars DRA 5.0 states that the Entry, Descent, and Landing (EDL) of a large payload on the surface of Mars remains a critical challenge for the human exploration of Mars [6]. It also states the issues associated with heat shields for a Mars atmospheric entry. Table 1 displays a comparative argument between aerocapture and propulsive capture. Figure 1 shows the orbital data for the Martian moons.

Table 1: Aerocapture vs. Propulsive Capture [6]

Question	Should the atmosphere of Mars be used to capture mission assets into orbit (aerocapture) or propulsive capture?
Recommendation	**Retain aerocapture for Mars cargo elements**
Notes	• Benefit of aerocapture is dependent on the interplanetary propulsion used (If NTR is used, the issue becomes one of risk. If chemical is used, aerocapture was considered enabling) • Aerocapture for the crew transfer vehicle was eliminated from consideration due to the physical size of that element
Notable Advantages of using Aerocapture for Mars Orbit Insertion	• Aerocapture reduces total architecture mass • Less architecture sensitivity to changes in payload mass • Minimal thermal protection system impacts. Both heat rate (factor of 3) and heat load (factor of 2) are less than those that will be experienced for Orion Earth return • Aerocapture guidance techniques are subsets of Orion skip trajectories
Notable Disadvantages	• Dual use of TPS (aerocapture followed by EDL) increases overall risk • Heat rejection and thermal load on primary structure yet to be assessed and will add mass and complexity



	Phobos	Deimos
Semi-major axis	9,376 km	23,458 km
Eccentricity	0.0151	0.0002
Inclination	1.075°	1.788°
Orbital period	7 hr 39 min	1 day 6 hr

Orbital parameters of Phobos and Deimos

Figure 1: Orbits of the Mars Moons [7]

The advantages of using a heat shield for aerobraking or aerocapture at Mars when using chemical propulsion are substantial. Aerocapture and aerobraking orbits at Mars are displayed in Figure 2. Typically a spacecraft would enter on an initial orbit with the use of aerobraking after an interplanetary transfer from Earth, with a periapsis close to Mars' atmosphere and an arbitrarily high apoapsis. This staging orbit can then be used to reference Phobos and Deimos transfer propulsion requirements.

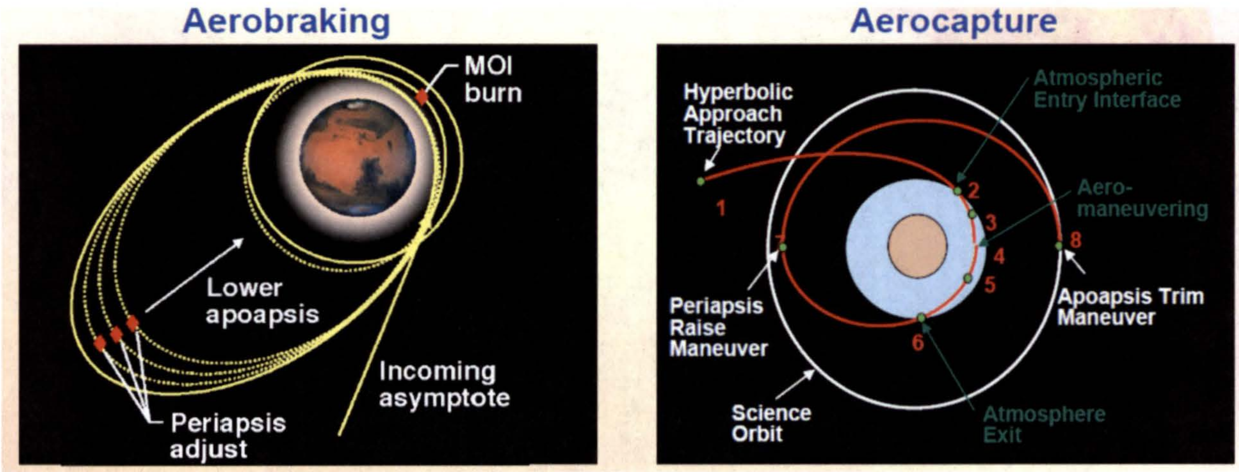


Figure 2: Aerobraking & Aerocapture at Mars [6]

By using a heat shield as the spacecraft approaches Mars to aerocapture into a staging orbit, the delta-v (Δv) for Mars capture can be reduced substantially. For example the Mars Global Surveyor (MGS) spacecraft was launched with a mission Δv deficit of nearly 1250 m/s. This reduction in the spacecraft propulsive capability was necessary in order to permit the development of a spacecraft design that satisfied the payload capabilities of the Delta II launch vehicle during the Earth-to-Mars ballistic launch opportunities of November 1996. The inherent mission velocity deficit was overcome by aerobraking to successfully establish the desired mapping orbit at Mars, [8]. Figure 3 shows aerocapture mass benefits from NASA Mars DRA 5.0.

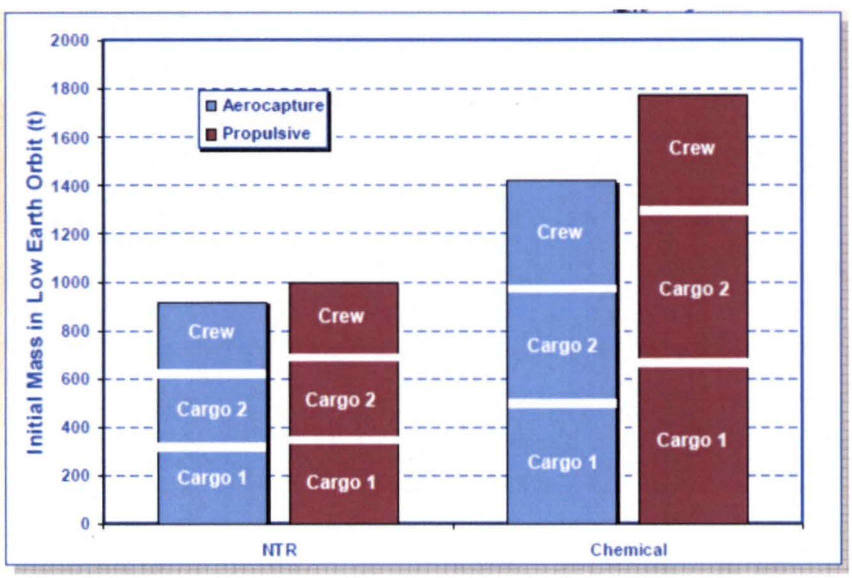


Figure 3. NASA Mars DRA 5.0 Aerocapture Benefits.

Similarly, for human missions, the initial mass in Low Earth Orbit can be reduced by using aerocapture as shown above. However the heat shield required to perform this aerocapture

sequence, is large and heavy. Table 2 shows that in Mars DRA 5.0, the total mass of the aeroshell structure and thermal protection system (TPS) is about 41 t, or 37% of the total in-orbit mass at Mars. Assuming a leverage (or gear) ratio of 6:1, the launch from Earth of a spacecraft without this heat shield can result in a savings of 246 t launched from Earth. A 57 t lander described in Mars DRA 5.0 (Table 2) would normally require a total of 341 t be launched to LEO (6:1); this can be reduced to a 95 t Inserted Mass in Low Earth Orbit (IMLEO) by using an in-situ fabricated heat shield. In that scenario, the 57 t lander and its transportation system can be launched by one heavy lift 100 t launcher, albeit with some additional operational complexity and risk. These calculations are approximations but serve to illustrate the benefits of the in-situ heat shield architecture. Further details will be calculated in future work. Table 2 gives the DRA 5.0 Mars EDL masses and parameters.

Table 2: NASA DRA 5.0 Mars EDL Characteristics [6]

EDL Mass Summary

Orbit Mass	**110.2**	t
Deorbit Propellant	0.5	t
Entry Mass	**109.7**	t
Aeroshell Structure	22.5	t
Thermal Protection System	18.2	t
RCS Dry Mass	1.0	t
RCS Propellant	1.2	t
Terminal Descent Propellant	10.1	t
Landed Mass	**56.8**	t
Dry Descent Stage	16.4	t
Payload Mass	**40.4**	t

EDL System Characteristics

Deorbit Delta-v	15	m/s
Balistic Coefficient	471	kg/m2
Descent Delta-v	595	m/s
Max Heat Rate	131	W/cm2
Total Heat Load	172	MJ/m2
Altitude Engine Initiation	1,350	m
Mach @ Engine Initiation	2.29	
Time of Flight	486	sec
Time at Constant g's	134	sec
Engine T/W	161	N/kg

This NASA Mars DRA 5.0 study assumed a 10-m diameter by 30-m long dual use shroud that can also be used as an ellipsled aeroshell for EDL. The TPS used was phenolic impregnated carbon ablator (PICA) and LI2200 that accounted for heating rates of 462 W/cm^2 during the aerocapture phase. Other options such as large supersonic parachutes and inflatable aerodynamic decelerators were also considered but not deemed mature technology. These options all point to large drag devices that can survive high heat fluxes and temperatures. The drag devices are constrained by the launch vehicle shroud diameter and the launch mass. Heat shield shapes can also vary [9].

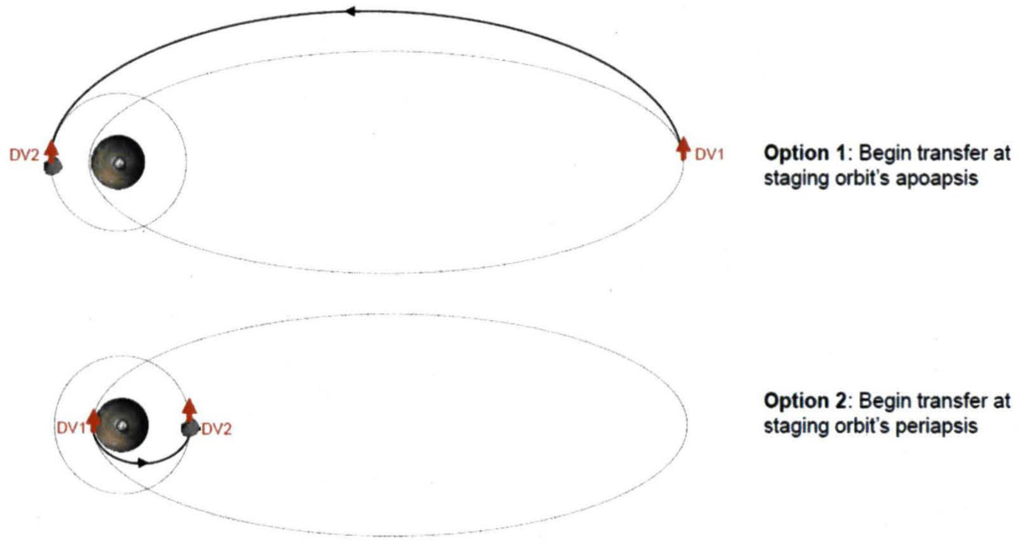
If a heat shield is fabricated in situ using local materials such as regolith from Phobos or Deimos, then the launch shroud size and the mass are no longer restricting parameters, and this constitutes *a "game-changing" solution for heat shields used for Mars aerocapture*. For the cost of the mass of the manufacturing equipment on Phobos/Deimos and the spacecraft and propellant required to transport the heat shield from there to a low Mars staging orbit, then large mass savings can be achieved. Since Phobos has a lower orbit and requires the lowest delta v if an aerobrake to a low Mars staging orbit is used, Phobos was chosen as the nominal location to fabricate in-situ Mars EDL heat shields. As long as the total mass required for making and transporting these in-situ

derived heat shields is less than 41 t, then a net mass savings can be achieved for a typical Mars mission, with corresponding cost savings as well. The regolith properties of Phobos are not known yet but it is assumed that they will have silicates and other minerals present which have been pulverized by meteoroid impact and space weathering to be similar to lunar regolith, which was used as a baseline material in this study.

This scenario involves the manufacturing of a Mars aerocapture heat shield on a Martian moon and its mating and assembly onto the incoming spacecraft prior to Mars aerocapture where it is needed, or the spacecraft will be destroyed. It is proposed that there will be two heat shields: one for Mars aerocapture, and one for Mars EDL. The Mars aerocapture heat shield will be fabricated on Phobos and then transported via a solar electric propulsion (SEP) tug to intercept the Mars-bound spacecraft. The SEP tug would pre-deploy via a Hohmann transfer into a staging orbit around the sun, where it would stay until the Mars-bound spacecraft intercepts it and execute a rendezvous. Once the heat shield is attached to the Mars spacecraft, then it is ready for aerocapture. Further work will be done on these orbital dynamics in a future work effort. Figure 4 shows Phobos and Deimos transfer orbits to a low Mars orbit (LMO) for entry insertion.

In the second stage of the journey (from Low Mars Orbit to the Mars surface), a new EDL heat shield will be used, and the Mars aerocapture heat shield will be jettisoned and replaced with a new heat shield to provide the highest quality and most reliable EDL capability. This will also reduce the amount of heat shield mass that has to be transported by the SEP tug for a deep space rendezvous. The Δv required to travel from Phobos to a Low Mars staging orbit is approximately 538 m/s with an aerobrake and 845 m/s without an aerobrake. Figure 5 shows an artist's concept of a regolith-derived heat shield protecting a payload as it enters the Martian atmosphere.

Our experimental proof of concept work shown in this report below, has identified a candidate material made from lunar regolith simulant JSC-1A that can function well as a heat shield material. A notional elliptical heat shield is shown in Fig. 6. The heat shield approximate mass has been calculated to be 98 t at the beginning of the ablative deceleration, with a decreasing mass during aerobraking. Further analysis needs to be done on ablation rates in a Phase 2 project, but this is a representative scenario for the purposes of estimation to show that a regolith heat shield could be very effective by ablating its mass away, without incurring a large mass penalty (even though it weighs twice as much as the Mars DRA 5.0 TPS) since it does not have to be launched and transported out of Earth's gravity well.



Propulsive ΔV	Option 1	Option 2	Option 2 with aerobrake
Phobos	845 m/s	1171 m/s	538 m/s
Deimos	604 m/s	888 m/s	651 m/s

Min ΔV option for Phobos if aerobrake is not available

Min ΔV option for Deimos regardless of aerobrake capability

Min ΔV option for Phobos if aerobrake is available

ΔV requirements by transfer option

Figure 4: Phobos & Deimos Transfer Orbits [7]

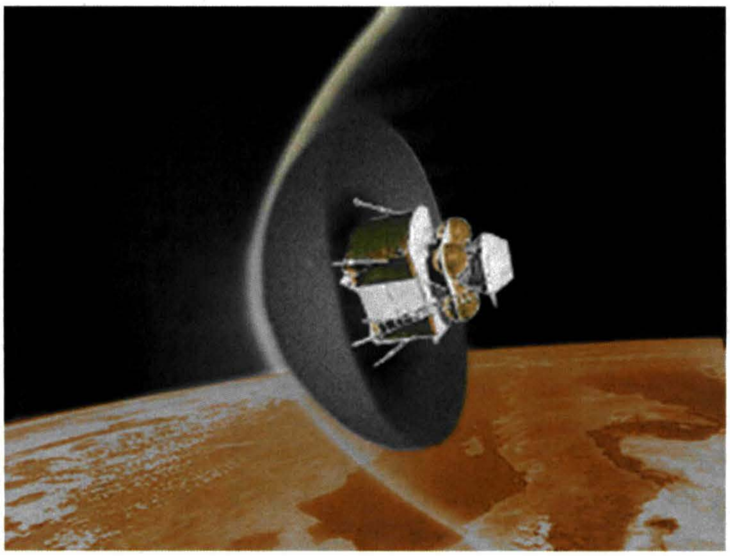


Figure 5. Artist's concept of a regolith derived heat shield entering the Martian atmosphere.

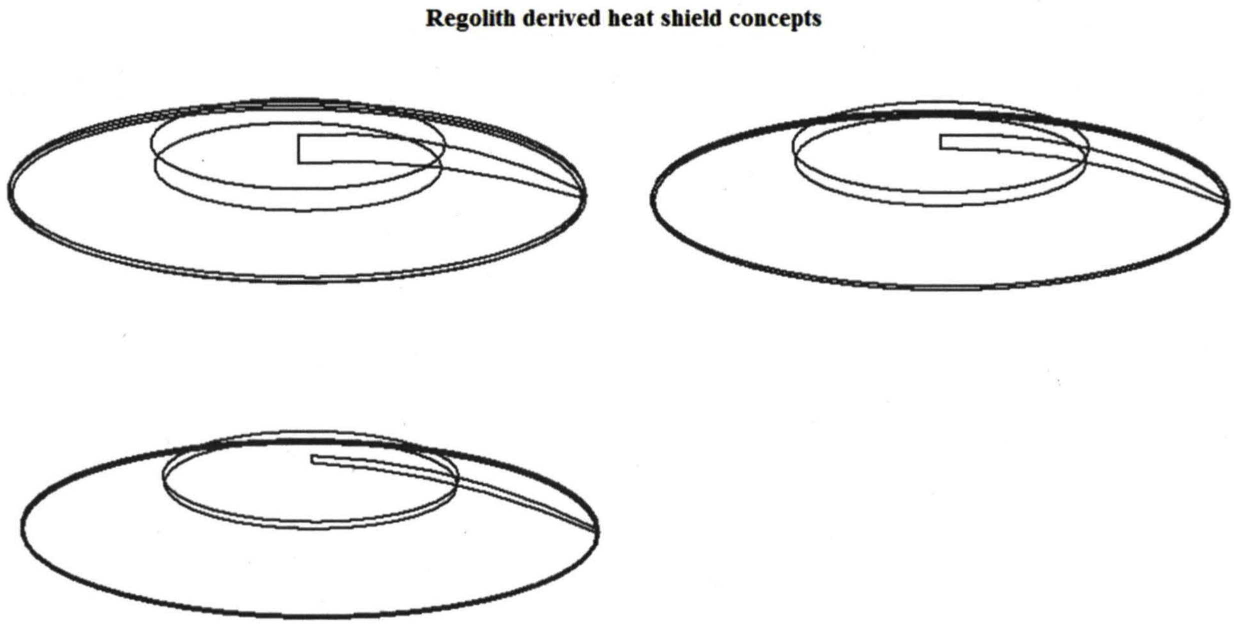


Figure 6: Blended Conic Elliptical Heat Shield Notional Concepts.

In this study it is assumed that the propellant is transported from Earth (with zero boil-off technologies) to provide the EDL heat shield transfers between Phobos and the Mars staging orbit, but it is possible that there will be in-situ water found either on Phobos itself or a nearby asteroid or comet, so that in-situ produced propellant in the form of hydrogen and oxygen could also be used eventually. The amount of hydrogen/oxygen propellant needed for a heat shield (mass = 98 t) delivery trip from Phobos to Low Mars orbit (Δv = 538 m/s), was calculated to be 12.7 t. The total mass of the transfer vehicle would then be approximately 14 t. We assumed that the EDL heat shield itself to have a diameter of 20-25 m [5] for a human scale mission with a lander mass of 50-60 t. With a gear ratio of approximately 6:1 then the total mass launched to LEO if propellant is brought from Earth would be 84 t. This results in an approximate mass savings of 246 t – 84 t = 162 t IMLEO. However, the dry mass of a Phobos to Trans-Mars Solar Electric Propulsion (SEP) tug (using Phobos In-situ derived H_2) would have to be subtracted from this since the Mars aerocapture cannot be performed without the delivery and rendezvous of an in-situ derived heat shield before arriving at Mars. Assuming the mass of the SEP is 13 t, then the transfer stage mass required to deliver it to Phobos would amount to 28.5 t (using Delta V= 4,845 m/s). So that means the total approximate IMLEO would be 84 t + 28.5 t + 13 t= 125.5 or about 126 t. The resultant mass savings IMLEO are now reduced to 246 t – 126 t = 120 t. Since the in situ fabrication equipment (5,000 kg) also has to be transported to Phobos at a IMLEO cost of 16 t, this *results in final savings on the first mission of 104 t or $520 M at $5,000/kg launch costs to LEO. Increasing savings on subsequent missions can be expected since the in situ fabrication equipment and SEP would already be in place on Phobos.*

In addition the heat shield fabricated at Phobos may weigh more than 98 t to add a safety margin to the ablation rate and in-situ fabrication equipment must also be transported to Mars to make the heat shield there robotically. Alternatively, the fabrication of a regolith-derived heat shield at Phobos could be even more viable via ISRU to make the propellant on the surface, and avoid the penalty of transporting the propellant from LEO. If water is available on Phobos then it would be possible to mine and process the water ice and fabricate the heat shield at a cost of about 10,000 kg of ISRU equipment (estimated) on Phobos which amounts to 68.5 t IMLEO, and this mass can be amortized over the lifetime of the architecture. Even if water is not present or not accessible, ISRU processes that extract oxygen from metal oxides that constitute the regolith would be employed to generate oxygen as a monopropellant. Further information about the Phobos regolith is needed to confirm the viability of the ISRU propellant solution.

We can envision repeated use of the architecture if the propellant could be made in situ at Phobos after the initial investment of transporting the mining robots and the ISRU propellant production plant; the full 246 t could then be saved on each subsequent mission resulting in a cost savings of $1.23 B each mission.

However, the separate Mars EDL heat shield must also be considered and that will also need a solar electric propulsion tug and propellants. In addition, transfer stages will have to be replaced over time. Assuming the SEP propellant can be created on Phobos using ISRU (e.g. H_2) then the amortized cost of propellants is negligible and only the cost of the amortized SEP stage must be considered. In addition, it is assumed that the same SEP used for Mars Insertion rendezvous is re-used. The completion of such trades and the viability of final solutions will also be the focus in a future study phase. Phobos and Deimos Δv budgets are shown in Fig. 7.

ΔV chart of optimal transfers

From \ To	Staging orbit	Phobos	Deimos
Staging orbit	-	540 m/s (w/ aero) or 848 m/s (w/o aero)	604 m/s
Phobos	848 m/s	-	748 m/s
Deimos	604 m/s	748 m/s	-

Example mission ΔV budgets

	Round-trip ΔV from staging orbit	
	w/ aerobrake	**w/o aerobrake**
Phobos	1388 m/s	1696 m/s
Deimos	1208 m/s	1208 m/s
Phobos & Deimos	1892 m/s	2200 m/s

Figure 7: Phobos & Deimos Delta V Budgets [7].

The fabrication of a Mars EDL heat shield also enables re-usability of a Mars Ascent–Descent vehicle fueled by propellants made via ISRU on the Martian surface (Methane/Oxygen). By replacing the heat shield on the Ascent-Descent vehicle when it returns from the Mars surface, a "Taxi" reusable architecture is possible, which can dramatically lower the cost of re-supplying the Mars Outpost, and once proven, eventually it can also ferry crew up and down from the Mars surface in a routine, affordable fashion.

The recent Mars Science Lander (MSL) mission has reached the limits of the current EDL technology set with very limited extension available:

- 0.9 t mass
- Largest Aeroshell ever flown (4.5 m) ever flown
- Largest ballistic coefficient (140+ kg/m^2)
- Highest Heat Rate (250 W/m^2) using PICA TPS
- Largest supersonic disk-gap-band parachute ever flown (21.5 m)
- Parachute was deployed at the highest Mach number (2.2)
- 10 km radius landing uncertainty ellipse

Figure 8 shows the MSL 4.5 m diameter heat shield. The estimated extensibility of this parachute EDL system is at most 2 t. A robotic Mars sample Return Mission will most likely require 1-3 t landed mass and a human-class lander would require 40-60 t of landed mass. These

facts indicate that a new type of Entry, Descent & Landing system will be required for human-class Mars missions [5].

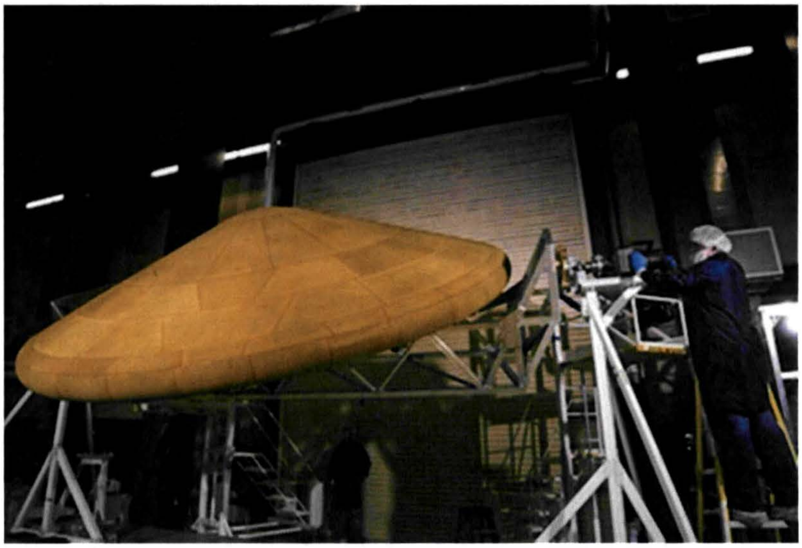


Figure 8: Mars Science Lab 4.5 m Diameter Heat Shield.

A landed spacecraft on the Martian surface launches a crew on its journey home to Earth carrying mineral, ice and atmospheric samples. After having spent over a year on the surface, the craft propulsion is propelled by methane and oxygen produced in situ. The spacecraft has been designed as a structure of minimal surface mass to ensure safe launch of its payload into Martian orbit and rendezvous with a Mars Orbiter. Once docked at the Orbiter, the spacecraft receives additional supplies, shielding and power generation equipment and a critical massive element it did not need on the Martian surface: a third heat shield for its Earth return. The heat shields are fabricated on one of the two orbiting moons of Mars, Phobos and Deimos and transported to the Orbiter that provides the mating platform. Once mated with its shield, the crew-transport journeys home to Earth. The heat shield is designed as an aerocapture shield to insert the craft into a low Earth orbit from which crew and cargo vehicles will descend to the ground and leave the space transport in orbit awaiting another journey to Mars. The heat shield could also be designed for a direct Earth entry trajectory, but this should be done only when a high confidence in the heat shields abilities have been proven. Figure 9 shows orbits and data for Mars opposition- and conjunction- class missions.

The automated and man-tended Fabrication Outposts at Phobos or Deimos can also fabricate heat shields for a variety of craft other than the Mars-bound and Earth-bound crew transports; the crew aboard the Mars Orbiter can conduct routine experiments of the upper atmosphere of Mars using probes designed and fabricated on-board and released with Mars entry heat shields fabricated at the orbital outposts. Some of these Mars entry craft may deploy dirigible aircraft high in the atmospheres while others bring supplies and spare parts to the surface crews. Both the

Mars-Earth and the cis-Mars transportation architectures are made possible by the in-situ capability of heat shield fabrication.

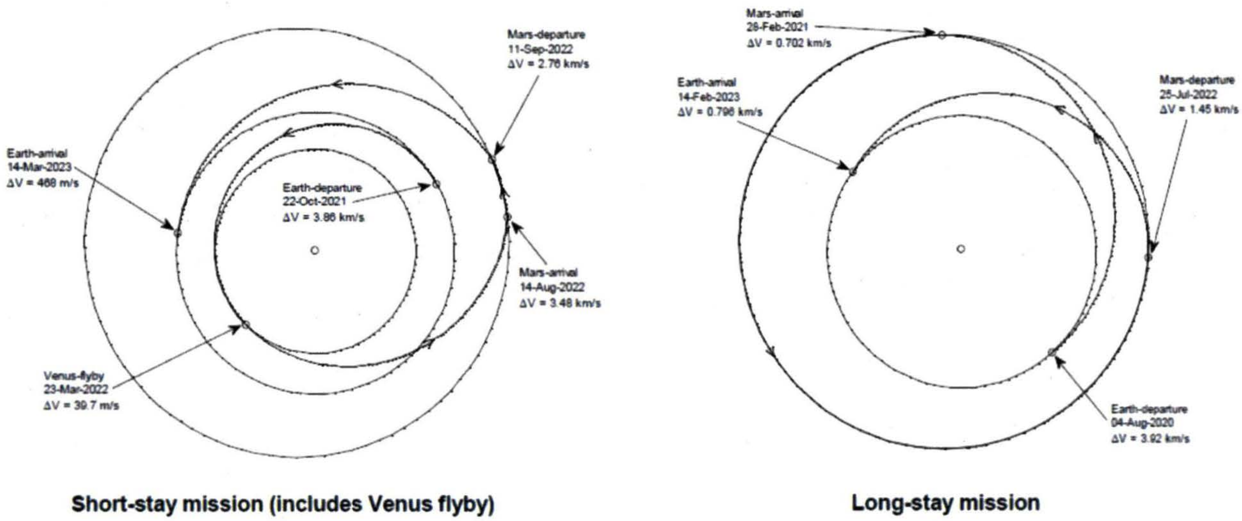


Figure 9: Mars Opposition and Conjunction Class Missions [7]

The analysis detailed above shows that a regolith-derived heat shield system could have a total IMLEO mass of 126 t (Phobos to Mars staging orbit) of which 5 t is required at Phobos for heat shield in-situ fabrication equipment. The rest of the mission is a Low Mars orbit to Mars surface EDL. This results in a mass savings of 104 t IMLEO to the crewed Mars transportation mission architecture in the initial Mars outbound journey. Additional savings would result if regolith heat shields would also be fabricated in-situ for the Earth return; however, since such shield would be smaller (e.g. a 5 m diameter Orion-class), then the risk trades may dictate using a pre-integrated heat shield to avoid in-space assembly (see Fig. 10). The maximum potential IMLEO mass savings are incurred by avoiding the launch of a Mars capture heat shield as well as the Earth return heat shield.

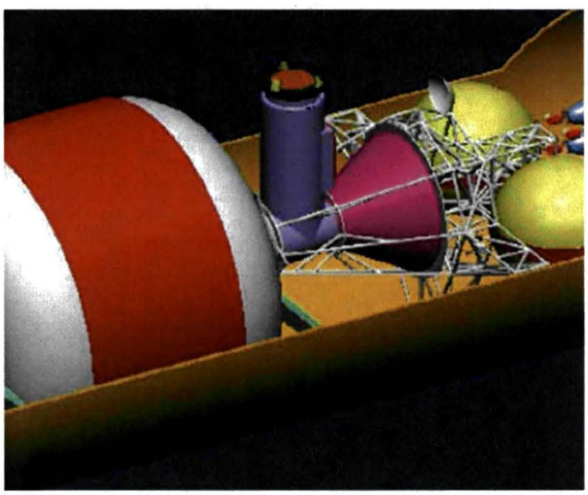


Figure 10: Return Capsule (purple element) Embedded in Structure: Hard to Access [5].

B. *Architecture II – Lunar Heat Shield Fabrication facility for Moon-Earth returns and Moon-Mars missions*

A lunar fabrication facility for heat shields on the surface of or orbiting the Moon may serve outbound transport spacecraft that need an aerocapture shield upon arrival at Mars. The servicing of spacecraft returning from Mars inbound to Earth in this scenario is unlikely as mission managers may choose not to take the risk of coming to Earth vicinity at interplanetary speeds without a shield.

The presence of robotic assets on the lunar surface is part of other architectures currently proposed and we propose to leverage and extend these capabilities to obtain the regolith needed for heat shield fabrication. The alternative options of on-surface and in-orbit fabrication are being studied and traded to best service crewed and cargo spacecraft leaving the Moon bound for Earth. The availability of a heat shield fabrication asset at the Moon would change any mission designed to shuttle between the lunar surface and Earth: in fact, such spacecraft could be launched from Earth without any heat shield since it only needs it during the final minutes of its mission during Earth EDL. We are studying the implications of such scenario on mass budgets at launch, lunar orbit insertion, lunar landings and operations and lunar launch.

Two mission architecture option hypotheses for the use of Lunar-made heat shields were considered:

1) *Fabricate a heat shield on the Moon, outfit it to an orbiting spacecraft and use it to aerobrake the spacecraft returning to Earth orbit or Earth Direct Entry (for sample return, mining ore, science payloads, and crew).*
2) *Fabricate a heat shield on the moon, deliver it to Low Earth Orbit (LEO) and assemble it to an outbound Mars Spacecraft for aerobraking or aerocapture at Mars.*

In the first case the reference used is the NASA Orion crew capsule heat shield (Figure 11), which is assumed to have a TPS mass fraction of 14%, similar to Apollo [10]. With a total estimated mass for the Orion capsule of 8,913 kg, this amounts to a **heat shield mass of 1,247 kg.** A notional design is shown in Figure 12.

Calculations using Tsiolkovsky' s rocket equation using a Δv of 4.04 km/s and an I_{sp} of 450 s yield an associated required propulsion capability of propellant and spacecraft weighing 2,834 kg in order to deliver this heat shield to LLO. An additional 474 kg of wet propulsion stage mass is needed to return the heat shield to LEO using aerobraking. This amounts to a total mass of **3,308 kg initial mass launched to LEO** required **to transport** the heat shield round trip to the Moon from LEO. **The total terrestrially manufactured heat shield system mass in LEO is** then 1,247kg + 3,308 kg = **4,555 kg.**

Figure 11: 5 m Diameter Ablator Heat Shield for NASA Orion Capsule.

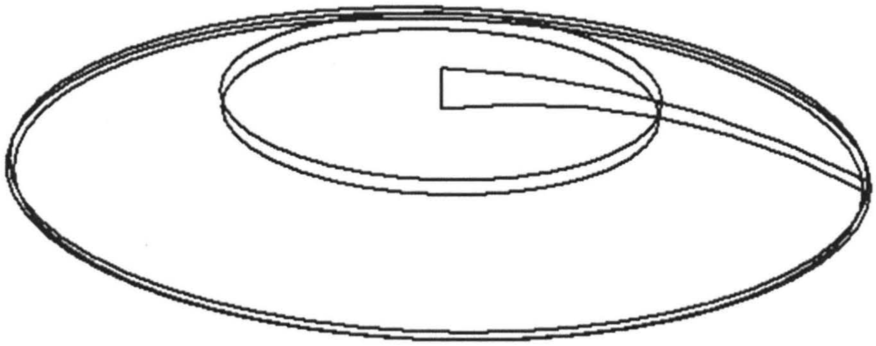


Figure 12: Notional Heat Shield Concept

If a regolith derived heat shield were used, then the mass of a shield is 2,300 kg. In addition, we must account for the amortized mass of the heat shield fabrication equipment and propellant production equipment delivered to the Moon. In this case it was assumed to be 5,000 kg with an amortized **250 kg** (20 missions) landed mass on the Moon per mission. This amounts to **a lunar ISRU surface delivery wet transfer stage mass of 832 kg** in LEO per flight (Delta –V = 6.14 km/s). The total equivalent mass required in LEO for ISRU propellant and robotic fabrication equipment and its delivery to the surface of the moon is then **832 kg + 250 kg = 1082 kg per flight**. The dry mass of the ascent stage/LEO transfer stage required for transporting the heat shield from the surface of the Moon to LEO was calculated to be **243 kg**, and the wet transfer stage required to deliver the ascent stage/LEO transfer stage there is **808 kg** resulting in an **IMLEO of 808 kg + 243 kg =1,051 kg.**

The LEO system mass for a Lunar ISRU derived heat shield system is only the mass of the amortized ISRU equipment, ascent stage and their transportation stages mass. The ISRU

equipment on the Moon uses solar energy that is free and local resources such as water ice, which is also free so that the mass and propellant for launching the heat shield from the surface of the Moon does not have the penalty of traveling through the Earth's gravity well to LEO. Therefore **the regolith heat shield system initial mass in LEO** (IMLEO) can be summed up to be 1,082 kg + 1,051 = **2,133 kg per mission.**

The IMLEO mass savings per Lunar orbital mission from using an ISRU regolith derived heat shield is then ***4,555 kg – 2,133 kg = 2,422 kg.*** *At a LEO launch cost of* ***$5,000/kg*** *this will result in a potential* ***cost savings of $12.1 M per mission****, so over a* ***20 mission campaign, cost savings would be $242.2 M.***

These kinds of cost savings could make a lunar commercial venture profitable or reduce the costs to the point where economies of scale help the business case. Further work is needed to do sensitivity studies and account for additional surface systems support equipment but this example illustrates the potential benefits.

In the second examined case we propose to fabricate a heat shield on the Moon, deliver it to LEO and assemble it to an outbound Mars spacecraft for aerobraking or aerocapture at Mars.

Using NASA Mars DRA 5.0 as a reference, the TPS and associated heat shield structure is assumed to be 41 t at a Mars staging orbit. With a gear ratio of 6:1 for LEO to Mars (as part of a 57 t lander)[10], then the IMLEO is 246 t; however a fair comparison must consider that the IMLEO of the heat shield itself is 41 t, so the lunar derived heat shield IMLEO must be less than 41 t to be viable.

The same heat shield system weighing 41 t could be fabricated on the Moon and then transported to LEO, therefore saving the propellant required to launch it from Earth to LEO. If 10,000 kg of ISRU equipment is assumed to be required on the Moon's surface to fabricate the Mars heat shield robotically and produce propellant to launch it from the Moon, and then this equipment is amortized over 10 Mars missions, and that amounts to the equivalent mass of 1,000 kg delivered to the Moon per mission. This would require 3,326 kg of wet mass for the transfer stage to the lunar surface per mission and the 1,000 kg ISRU payload resulting in 4,326 kg IMLEO per mission for delivering the ISRU equipment.

The ascent stage and the lunar orbit to LEO transfer stage must then be transported to the lunar surface requiring a wet transfer stage (LEO to Lunar Surface) of mass 58,828 kg. The ascent stage and transfer stage have a combined dry mass of 17,685 kg and are then filled with lunar derived ISRU propellants. The lunar regolith heat shield for Mars entry weighing 98 t would have to be transported from the lunar surface to LEO. It is assumed here that the heat shield does its own Earth aerobraking, so that a separate LEO aerobraking heat shield can be avoided. Inspections in LEO would then certify it for use in a Mars mission. Alternatively a nested heat shield could be produced where the Earth heat shield protects the fresh Mars capture heat shield, but this would require more payload mass to be transported from the Moon to the Earth.

By adding up the various components, then ISRU 4,326 kg + Dry LEO Ascent Stage & Dry Moon to LEO: Wet Transfer Stage from LEO 58,828 kg + Lunar Ascent & Moon to LEO Transfer Stage Dry 17,685 kg = **80,839 kg total IMLEO per mission for the regolith heat shield for Mars aerocapture.**

Comparing this to the IMLEO 41,000 kg baseline heat shield and transportation system mass that would be needed if it were not fabricated on the Moon, this results in a **IMLEO deficit of 41,000 kg – 80,839 kg = -39,839 kg.**

This analysis has shown that the additional Δv burden imposed by the gravity well of the Moon and the added mass of using the heavier regolith as TPS means that the lunar-made regolith heat shield for Mars is not a viable solution. Launching a heat shield from Earth and transporting it directly to Mars without a lunar fabrication intermediate step can minimize the IMLEO.

C. *Architecture III: In Situ Heat Shield Fabrication at an Asteroid for Earth-bound Spacecraft*

The mining of asteroids for resources of value is now being evaluated seriously thanks to the rapid advances of robotics and new launch capabilities. Several corporations funded privately have emerged to invest in the search of target asteroids and technologies needed to explore and mine them. The economic viability of such endeavors is still very much in question because so many known and unknown risks exist; our current knowledge of these objects remains fragmented and superficial and does not yet enable us to develop adequate technologies for landing on, mining and processing asteroids.

Nevertheless, viable exploitation of asteroids for high value metals to return them to Earth where such resources will become too scarce or inaccessible in the future must exploit technologies that will reduce large expenditures. Equipment and spacecraft components for such missions will comprise the largest portion of the launch mass sent from Earth. The return of mined resources from space to Earth will require heat shields for robotic cargo shuttles that will bring the materials to either LEO or to Earth surface. If metal extraction and refining facilities are assembled at LEO, these cargo craft may conceivably use aerocapture heat shields to slow themselves from interplanetary travel velocities to LEO orbital speeds and rendezvous with the orbital processing facilities. Once metals are extracted and refined from asteroid ore, cargo shuttles will 'drop' from LEO to Earth surface with their precious payloads.

This mission architecture highlights the need for in situ Heat Shield Fabrication near the mined asteroid to manufacture and install the aerocapture shield for each cargo craft. The ore being transported by such craft to LEO will still contain a large mass portion of silicates since metal extraction facilities may not be constructed at the asteroid itself. Once the ore is processed and the silicate gangue is separated from the target metals, these silicates will be formed into single-use heat shields for the shuttle cargos to bring the metals to Earth surface. The arc jet tests we

performed established that such materials can be used to fulfill heat shield functions for Earth reentries.

If a typical baseline near-Earth asteroid Δv of 5.5 km/s is assumed, then the mass savings associated with an in-situ produced heat shield can be calculated. It is assumed that the payloads that can be brought back from an asteroid must be reasonable so that in the event of a loss of control, the mining ore spacecraft would burn up in the Earth's atmosphere and not pose a hazard to the population. Therefore a payload mass of 7.7 t was chosen (similar to Orion), which could be returned with a 5 m diameter heat shield made of asteroidal regolith attached to the transfer stage. Assuming that the 5 m heat shield has a mass of 1,247 kg as in the Orion lunar example above, then the return propellant for the ore with the heat shield payload would also have to be factored in (with the use of aerobraking at 50% Delta V savings [11]) and amounts to a mass cost of a wet transfer stage of 8,487 kg and in order to transfer this return stage with an Earth fabricated heat shield from LEO to the Asteroid, it would require a **LEO – Asteroid transfer stage of 26,558 kg IMLEO.** The total baseline comparison mass for a mission with Orion-type heat shield **returning 8.8 t of ore would then be LEO to Asteroid stage 26,558 kg + Asteroid to LEO aerocapture 8,487 kg + Heat Shield 1,247 kg = 36,292 kg IMLEO per mission.** At \$5,000 per kg launched to LEO this amounts to a cost of \$181.5 M, which means the market value of the ore must exceed \$23,566 / kg to produce a profit. Since the mining equipment delivery cost is not included here, the actual asteroid mining cost would be even higher. Since the current market price of platinum (one of the most valuable metals on the market) is approximately \$50,000 / kg, this means that a rough profit of \$26,434 / kg could be produced. With a LEO payload of 7,700 kg, this translates into potential profits of \$203.5 M per mining mission, although the cost of purification and return to the Earth's surface would still have to be subtracted from this margin. However this rough estimate shows that mining an asteroid for platinum is potentially lucrative.

However the heat shield can be fabricated at the asteroid instead. In this case the ISRU heat shield fabrication equipment would have to be transported to the asteroid and it is assumed that the propellant is brought from Earth, since a metallic-type asteroid may not have water ice present for ISRU propellant production. Alternatively, oxygen can still be produced from such asteroid by using ISRU processes that reduce the metal oxides in the ore such as Molten Regolith Electrolysis [12][13], hydrogen- [14] and carbon- reduction [15] of the regolith. These processing approaches would certainly be part of the metal extraction schemes to obtain the targeted metals. It thus makes sense that the oxygen released from the metals be harvested and used as a monopropellant in ion engines for example. The wet mass of the ISRU fabrication equipment transfer stage is 2,729 kg and it carries 5,000 kg of ISRU fabrication equipment so that the total IMLEO for this asteroidal heat shield fabrication capability would be transfer stage mass 13,642 kg + ISRU 5,000 kg = 18,642 kg IMLEO. If this was amortized over 50 missions it would be reduced to **372 kg per mission**.

In addition, the mined ore of 7,700 kg + 2,300 kg regolith heat shield would have to be transported back from the asteroid to LEO for processing at an orbital outpost prior to returning the high grade ore to the Earth as explained above. This would require a wet mass transfer stage of 9,521 kg and in order to transfer this return stage to the asteroid it would require a **LEO – Asteroid transfer stage of 25,980 kg IMLEO.** The total mass for an **asteroidal regolith-derived heat shield mission would then be 372 kg + 9,521 kg + 25,980 kg = 35,873 kg IMLEO per mission.**

By comparing these two cases it can be seen that **the net savings per mission amount to** 36,292 kg - 35,873 kg = **419 kg IMLEO**. The bulk of the mass being transported is propellant, and since the regolith derived heat shield is heavier, then the benefits of ISRU fabrication are somewhat more profitable (~$2M) versus sending an Earth fabricated heat shield directly, but the risk is higher. If an asteroid could be found with good ore and water ice for propellants then the benefits of ISRU would be far greater, since in-situ propellants could be used which would avoid sending the return journey propellant at substantial mass savings

If the asteroid did contain water for propellants or a nearby asteroid did, then the total IMLEO would be ISRU 745 kg + Dry Return Stage 865 kg + LEO to Asteroid Wet Transfer Stage 2,363 kg = 3,973 kg. ***By using ISRU propellants at the asteroid, the regolith-derived heat shield and ISRU become viable with a mass savings of 36,292 kg - 3,973 = 32,319 kg IMLEO that translates into $161.6 M per mission cost avoidance at $5,000/kg launched to LEO.***

Now the economic case becomes much more attractive, since 3,973 kg IMLEO costs $19.9 M to launch. This corresponds to a mining cost of $2,580 / kg. With an assumed market price of $50,000 /kg of platinum ore, for example, then the profit margin is $47,420/kg. With a LEO payload of 7,700 kg this yields rough earnings of $365 M in LEO per mission or $162 M higher than without ISRU propellants and in-situ heat shield fabrication.

III. Regolith-based Heat Shield In-Space Fabrication Concepts

The consolidation of granular regolith into monolithic solids can be achieved using several techniques that either fuse the grains by heat application (sintering, melting) or bind them through chemical reactions or added binders such as resins. Sintering represents a very attractive option as it does not require the importation of any chemical or binding agents and requires less thermal energy than melting regolith. Experimental work performed for this project shows that sintered regolith samples are less dense than binder-filled samples except for those that contain higher than 15% binder by volume. While materials with high binder content have lower densities and higher tensile strength they also require large amounts of imported compounds typically not found in situ in a native state. For this reason, we have focused on fabrication of

sintered regolith heat shields in the following feasibility assessment study of the architectures examined by this project.

Additive manufacturing technologies are selected here because they are uniquely designed to handle and process granular or powder materials such as regolith. They also lend themselves well to full automation and precise robotic execution that are essential for deep space operations. A novel technology that has recently emerged is Contour Crafting pioneered by Prof. B. Khoshnevis that received NIAC Phase I (2011) and II (2012) awards [16]. We have selected this technology for our architecture study because it allows rapid construction of large-scale structures using regolith through robotic construction. Contour Crafting incorporates the use of a material processing and delivery 'head' at the extremity of a robotic arm; this particular feature offers powerful versatility in preparing, processing and fusing added material to form a finish structure. In fact, operations can include the interchange of such attachments that perform sintering of layers of delivered regolith with concentrated photon beams (solar or laser) or deliver regolith mixed with binding compounds with heat or irradiation curing.

The creation of designed shapes for heat shields can be accomplished by different approaches on planetary surfaces depending on the diameter and overall size of heat shields. Molds can be brought with spacecraft in sections and assembled upon landing but the fabrication of molds is also feasible in situ using packed regolith. Topographical features such as craters or ground depressions provide first approximations of the desired shapes and save time and energy to excavate the corresponding regolith mass out of a flat surface. In situ sintering of layers of regolith to the desired thickness has been demonstrated in regolith molds. The cutting and milling of rock has been considered but the energy expense and the high density of the final product do not favor this approach.

A. Architecture I – Regolith Heat Shield Fabrication Feasibility at Martian Moons

The low flux density of solar radiation reaching the orbit of Mars limits its potential as an energy source for many space missions in the region. Small yet capable rovers have made very efficient use of their solar panels (Sojourner, MERs) but large landed craft such as Viking and Curiosity rely on radioisotope thermoelectric generators (RTG) for higher power density and reliability. The fabrication of regolith-based heat shields on Phobos requires such reliable power both for processing the regolith and to operate the infrastructure surface equipment. Autonomous rovers ferrying between excavation sites and hoppers that bring the granular material into a Contour Crafting robotic head would perform the excavation of regolith. These rovers can also create the surface mold used to build the shield as well as servicing the various components of the infrastructure.

The thermal energy needed to achieve sintering of the regolith would be accomplished by the use of lasers integrated in the Contour Crafting head since solar heating is not a viable option. 440 W lasers used successfully in regolith melting are used as initial element of this feasibility

assessment [15]. The selection of nuclear energy presents us with a limited list of devices of state of the art technology or under development. The Multi Mission Radioisotope Thermoelectric Generator (MMRTG) that powers the rover Curiosity on the Martian surface at present represents current RTG technology; each unit delivers ca. 110W_e for a mass of 11.5 kg. Each sintering laser above would require 8 MMRTG units when one accounts for conversion losses and power margins. Although this system configuration offers redundancy and relative mobility, it does not lend itself well to providing margins of power for the whole surface architecture. The next nuclear system considered is the Fission Surface Power (FSP) under development by NASA GRC and DOE [17]. In its full configuration, it would provide 40 kW_e on planetary surfaces and is the best option for our architecture in the Martian system. The FSP mass remains undefined at this point so we selected a notional 600 kg mass as reference in our study. This estimate is an extrapolation of previous reactors such as the SNAP-10 (USA) that flew in space in 1965 providing 0.65 kW_e with a mass of 435 kg and the Topaz series (USSR) flown in 1987 yielding 5 kW_e for 320 kg. We accounted for the positive impact of technology advances and the added mass represented by Brayton cycle turbines or piston engines, alternators and Stirling engines in the FSP while the older reactors used thermoelectric or thermionic elements for power conversion. An FSP energy source offers excess power for sintering additive manufacturing at different scales with varying heat shield diameters and the supporting surface infrastructure. The latter includes; excavating rovers with motorized anchoring to operate in reduced gravity ferrying regolith from selected sites to hoppers, regolith transport systems to sintering heads, the Contour Crafting system itself on anchored bases, and a power management and distribution (PMAD) system with beamed power capability. An additional system capable of lifting the fabricated shield from the surface, and secure it to a propulsion package is estimated.

Architecture I describes the needs of a variety of missions with different heat shield requirements. An Orion-class craft incoming from Earth to Mars may require a 5m-diameter EDL shield and a larger aerocapture shield that we modeled as a 20m-diameter shield for this initial study. The servicing of shuttlecraft between the Martian surface and orbiters and scientific craft launched from Mars orbiters to study the atmosphere or deliver payloads to the surface is a compelling scenario for regolith heat shields fabrication on Phobos. These small mission spacecraft equipped and prepared aboard a manned orbiter at Mars would require small shield size represented by a 1m-diameter shield scale in our calculations.

Our preliminary estimates show that 1.0 m-diameter regolith shields could be fabricated by a modest infrastructure on Phobos in the following scenario. It would include two small excavating rovers, one Contour Crafting system with two sintering lasers and robotic manipulators to ready and launch the shield. If one includes a propellant-making ISRU plant to electrolyze water found on Phobos or extract oxygen from silicates to fuel an ion engine, initial calculations show that the shield could be fabricated in 15-20 days with a surface infrastructure of 2 t requiring 12-13 kW_e. As pointed out earlier, this scale of operation can barely be sustained with MMRTG-type power, as it would require 77 of those units that in turn increases the landed mass. Alternatively,

the selection of FSP as a single power source provides high margins and enables modular expansion of the infrastructure to larger shield fabrication while keeping the landed mass between 2 and 5 t.

The fabrication of a 5m-diameter and a 20m-diameter shield using FSP would include the use of two to three Contour Crafting robotic heads respectively working symmetrically to build sintered material employing up to four sintering lasers in succession per head. As scale increases, additional excavation rovers and transport capability is added powered by a single FSP. Current order-of-magnitude calculations indicate that the largest infrastructure with three Contour Crafting robots and four rovers with a propellant ISRU plant would require ca. 35 kW_e. It is important to note that the fabrication and launch of a very large diameter heat shield is made possible by operations in reduced gravity (milli- or micro-G) that reduce the strain level on the sintered regolith part although much research and technology development needs to be done to realize the actual fabrication techniques in such environment.

B. Architecture II – Regolith Heat Shield Fabrication Feasibility on the Moon

The lunar surface environment offers significant advantages for fabrication when compared to the Martian moons. Its higher gravity field helps in mitigating granular material flow problems and facilitates thermal transport during heating processes such as sintering. It also provides more stability in large structure handling and roving/excavation operations. It offers a critical source of energy in the solar radiation. This enables a lunar architecture to perform direct thermal sintering by concentrated solar beam systems thus avoiding electrical to thermal conversion in lasers. Our order-of-magnitude calculations utilize the state-of-the-art Cassegrain solar concentrators used in current ISRU plant development as a reference [18]. Such systems provide ca. 6 kW of thermal energy to the regolith for a 34 kg hardware mass when using solar flux densities on the Moon. Our architecture thus includes Contour Crafting heads equipped with piped-in solar beam sintering. Based on the energy requirements and supply, a 1m-diameter shield can then be fabricated with one solar sintering fabricator head in 6 to 10 hours when regolith excavation and transport is taken into account. Similarly, 5 m- and 20 m-diameter shields can be achieved in 7 and 33 days respectively with the use of four and eight sintering beams in each case. These numbers reveal engineering challenges as the number of solar concentrators increase but the adaptability of the Contour Crafting approach offers potential solutions. While the sintering can be performed without electrical power, the rest of the surface infrastructure requires an electrical source. Solar panels augmented with energy storage can provide it on the lunar surface but the FSP remains also an attractive option. The MMRTG option proves to be too limited for the same reasons as previously treated in Architecture I at Phobos. The total architecture electrical power on the lunar surface ranges from 13 to 30 kW_e when propellant plants are included for a total mass ranging from 2 to 5 t.

The launch of heat shields from the lunar surface decreases the level of feasibility of the architecture as shown previously. It also poses challenges in their assembly and launch to LLO

or LEO; while 1m-diameter shields can be fitted in a transport, this is much more challenging for 5 m and 20 m-diameter items. This could require the fabrication of large regolith heat shields in sections or panels that would need to be fitted, assembled and fastened in orbit. This level of added complexity has not yet been evaluated in our work.

C. Architecture III – Regolith Heat Shield Fabrication Feasibility at Asteroid

The proposed infrastructure for regolith-based heat shield fabrication on an asteroid mirrors that described in Architecture I for Phobos. The same assumptions are applied here since Phobos is considered a captured asteroid in orbit around Mars. Additional challenges arise, as the transportation architecture would likely be less developed around an asteroid than in the cis-Mars system where orbiters, telecommunication assets and surface assets play supporting and enabling roles. The necessity to bring power sources such as FSP and multi-element surface infrastructure for shield fabrication to an asteroid assumes that the target is of very high value with water and mineral resources present.

IV. Heat Shield Formulations

Three types of fabrication methods for several heat shield formulations were investigated. One method is to sinter the native granular regolith to form a solid mass. Sintering is accomplished at temperatures high enough for the particles that make up the regolith to begin to fuse or stick together without fully melting. The resulting solid body has a lighter density compared to the density of a full melt. The second method to fabricate the heat shield is to use the post-processing hot regolith from an ISRU reactor. Several reactor designs involve processes to remove oxygen from regolith that result in a by-product stream of very hot if not melted material similar in composition to the original regolith. The ISRU by-product material stream can then be used to create sintered materials from the regolith also so this method was combined with the sintering method as one of the potential ways to effect sintering. We discarded the use of fully melted stream materials because their higher density, higher thermal conductivity and generally weaker mechanical strength once solidified make them less desirable candidates for heat shield applications. The third method is to bind the regolith particles together using a high-temperature RTV. In this case, the goal is achieve a ratio of binder to regolith low enough to limit the mass of binder needed while creating a mixture that can be cast in desired shapes and perform as heat shield components.

The density of the resulting heat shield material is of significant consequence to its viability. If the density is too light, the heat shield will not withstand the stresses of atmospheric entry. However, if the material is too dense, its thermal conductivity will be too high to adequately protect the vehicle or payload.

A. Sintering Method

Two regolith simulants were used in the sintering experiments. They are: JSC-1 Mars, JSC-1AC Lunar. These simulant materials were selected for their availability and on the basis of their representation of relevant characteristics of lunar and Martian regolith for this work. Although JSC-1 Mars is not considered a good simulant of any particular soil material found on Mars, it approximates the silicate and iron oxide content of such materials. JSC-1A is standard simulant produced to reflect the chemical and mineral composition of lunar basalts and JSC-1AC includes the wider particle size distribution that approximate these natural materials before they undergo any processing. Molds were made from Fondu Fyre™ (a high temperature ceramic material) to make approximately 6" x 8" x ½" tiles. Sintering was done in a Paragon furnace. This fabrication is shown in Fig. 13 [2].

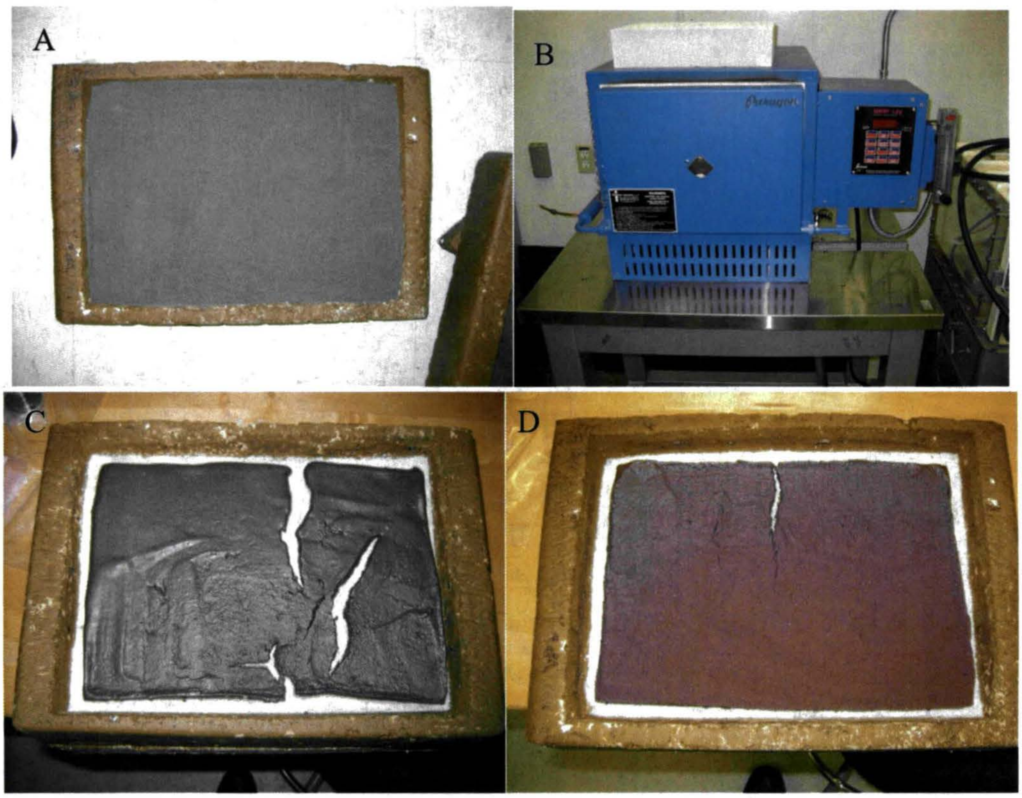


Figure 13. A. JSC-1AC regolith simulant in Fondu Fyre mold. B. Paragon furnace. C & D. Resulting sintered regolith. Reddish color of the lunar simulant is due to oxidation of iron during sintering in air.

Shrinkage and some cracking were noted on the resulting tiles. The sintering process was refined for temperature and time to produce the samples used in flame and arc jet testing.

B. RTV Binding

The RTV used is a two part RTV formulated to retain its elastomeric properties at high operating temperatures. It also retains flexibility to low temperatures. Several three inch diameter by ½" test samples were made using JSC-1A lunar and JSC-1 Mars regolith simulants. The samples were fabricated by adding an amount of the RTV to the regolith in various ratios. The lowest percent RTV formulations did not have sufficient cohesion after the RTV set and were disqualified. Two of these initial test samples are shown in Fig. 14.

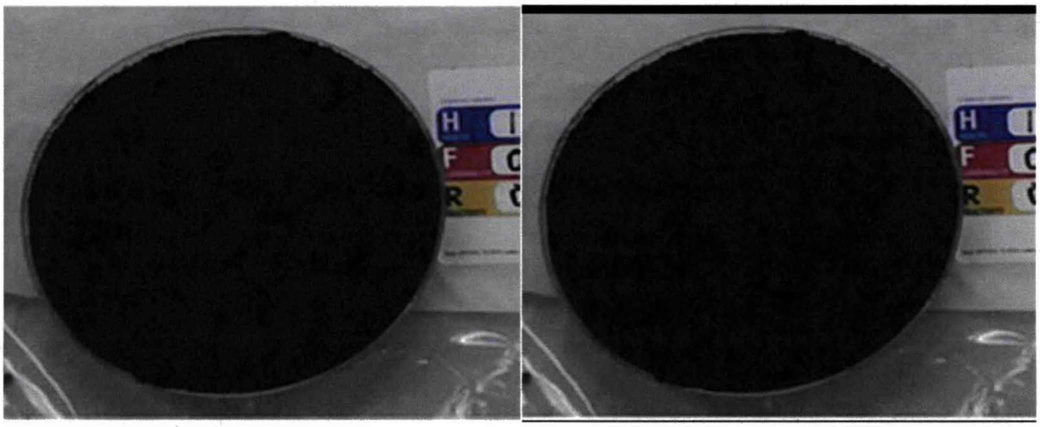


Figure 14. Initial RTV bound test tiles formed in a 3 inch diameter by ½" petri dish.

V. KSC Internal Evaluation and High Temperature Flame Testing

A. Test Sample Design

To determine which of the formulations are viable for heat shield use, it was decided to expose samples to an acetylene cutting torch pursuant to arc jet testing at ARC. To ensure a good fidelity test, a sample shape was designed based on similar 4" diameter by 2" thick samples designed for tests in the arc jet.

Two types of molds were fabricated to make these iso-q samples. For the sintered samples, Fondu Fyre™ molds were made. For the RTV bound samples, plastic molds were made from a polymer via 3D machining methods. An example of each of these molds is shown in Fig. 15.

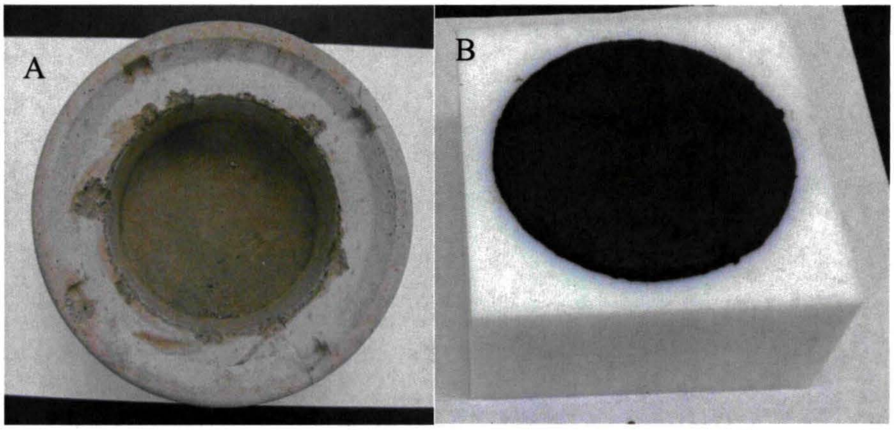


Figure 15. A. Fondu Fyre mold for sintered samples. B. Polymer mold showing a JSC-1 Mars simulant/RTV sample as it cures.

B. Initial Formulations

Two sintered and six RTV bound samples were fabricated for internal KSC flame testing and are listed in Table 3.

Table 3
Samples for KSC Internal Testing

Simulant	Composition
JSC-1AC Lunar	Sintered
JSC-1AC Lunar	Sintered
JSC-1A Lunar	RTV Bound
JSC-1A Lunar	RTV Bound
JSC-1A Lunar	RTV Bound
JSC-1 Mars	RTV Bound
JSC-1 Mars	RTV Bound
JSC-1 Mars	RTV Bound

C. High Temperature Flame Tests

To evaluate the viability of the initial test sample formulations, it was decided to expose the samples to the heat of an acetylene cutting torch for five minutes. This exposure represents the longest planned exposure in the Arc Jet Complex tests and approximates the heat load and some ablation that will be placed upon the samples in the Arc jet facility. The torch was positioned approximately six inches from the front surface of the sample. The flame temperature estimated at that distance from the torch nozzle is approximately 2200° C [21].

Each sample was placed horizontally on a large welding table supported by Fondu Fyre™ bricks. To measure rear temperature, a type K thermocouple was attached to be back side of each

sample. The output of the thermocouple was read via an Omega Omnical. To measure the front temperature, a Fluke model Ti 32 IR camera was used. While rear temperatures could be measured throughout the test, the Fluke IR camera was limited to readings up to 620°C so these readings were taken during the cooling period immediately following the five-minute torch impingement was completed. Each test run was recorded via a video camera. Front and rear temperatures were recorded via the above-mentioned devices for eight minutes after torch termination. The test set up is shown in Fig. 16.

In all of eight test runs, the maximum rear surface temperatures did not exceed acceptable levels. The maximum temperature was measured a few minutes after flame test termination. This indicates that no significant heat was transmitted to the rear of the samples during the test. Post-test condition of the front surfaces of the samples is shown in Fig. 17. An example of the temperature data is shown in Fig. 18.

A very dense sintered sample of JSC-1A Lunar simulant was provided by KSC QNA-ESC for comparison to the eight samples. This sample, approximately 2.25 inches in diameter by 1.0 inch thick, was sintered under a 30 psi Helium atmosphere. Test data could not be taken for this sample because the density of the sintered structure allowed too much heat to penetrate to the rear of the sample melting the thermocouple junction.

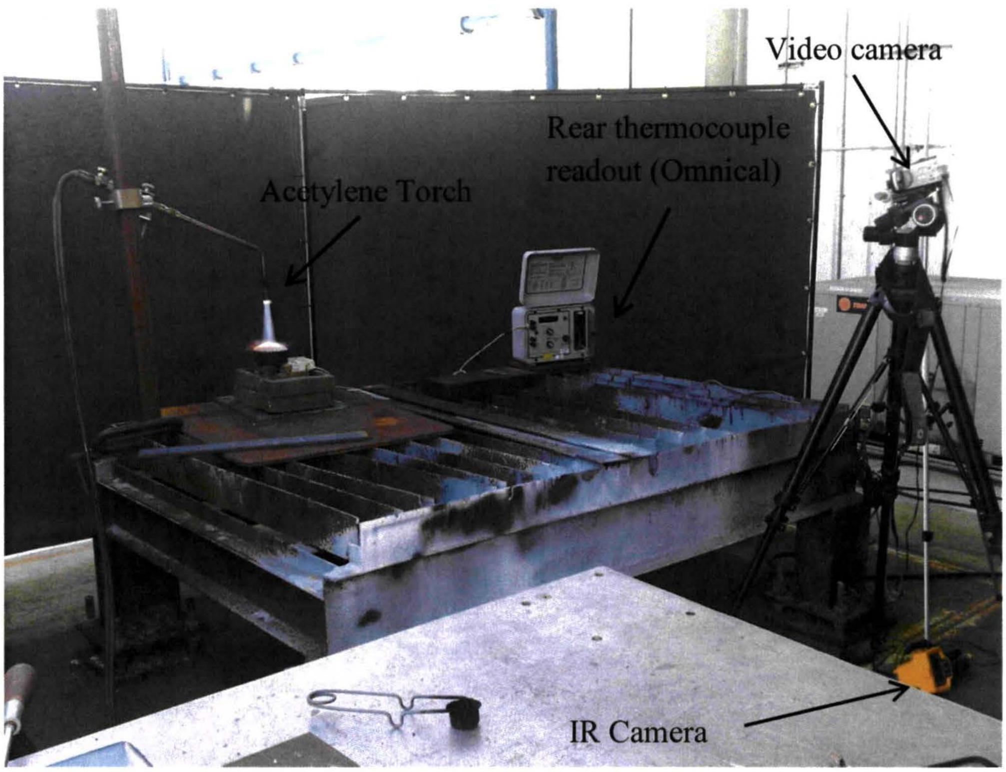


Figure 16. High temperature flame test set-up at the ESC Weld Shop (LETF) at KSC showing a heat shield material sample under test.

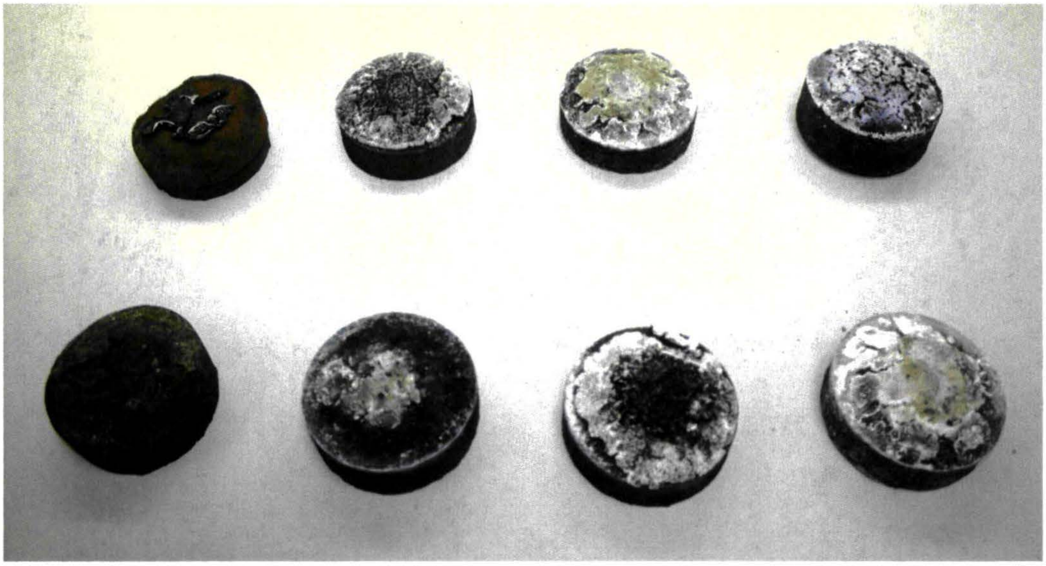


Figure 17. Post-test photo of the front surfaces of the eight flame test samples. The two JSC-1AC Lunar sintered samples are on the left-most column. The remaining samples on the top row are: RTV/JSC-1 Mars, RTV/JSC-1A Lunar, RTV/JSC-1 Mars. On the bottom row: RTV/JSC-1A Lunar, RTV/JSC-1 Mars, RTV/JSC-1A Lunar

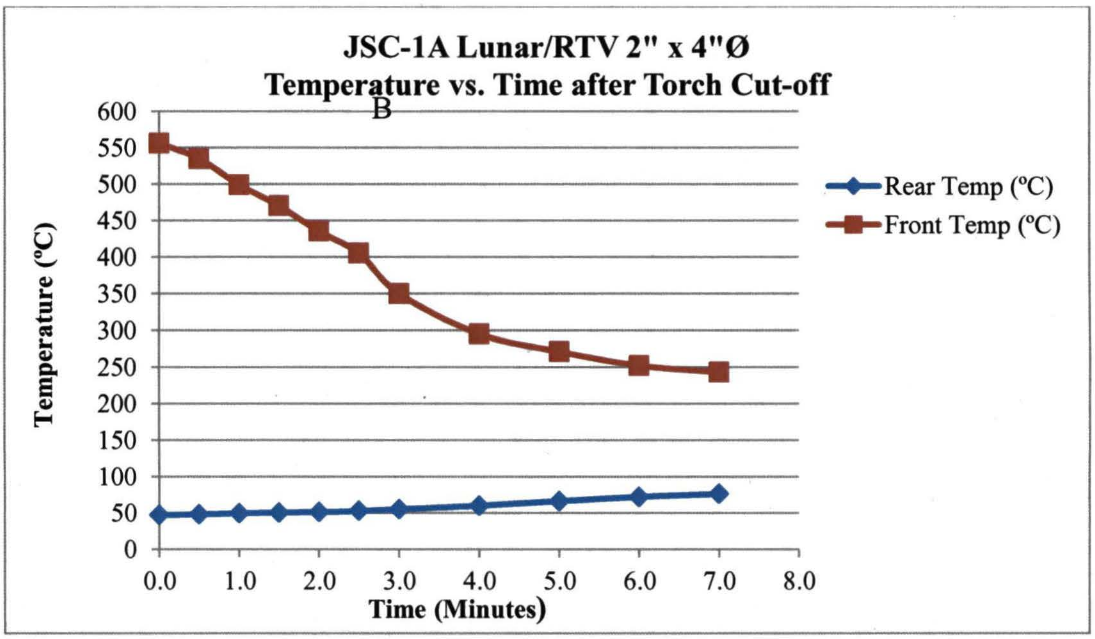


Figure 18. High Temperature Flame test temperature data acquired for the RTV/JSC-1A lunar sample. Measurements start at the beginning of the material's cooling period after termination of the exposure to the flame.

VI. Arc Jet Tests at Ames Research Center

The IHF Arc Jet test facility at ARC can expose test samples to a low pressure but very hot Argon and air plasma stream that models the thermal and dynamic conditions of atmospheric entry. Ten of the 4 inch-diameter by 2 inch-thickness samples were fabricated for this testing. Sintering of the JSC-1 Mars simulant resulted in cracking and excessive shrinkage so these samples were not used. The three sintered samples sent for arc jet testing were made from JSC-1AC Lunar simulant. The remaining seven samples were RTV bound formulations of both JSC-1A Lunar and JSC-1 Mars regolith simulants. Table 4 summarizes the selection of the ten Arc jet samples tested at the IHF Arc Jet Facility.

An Aluminum back plate was designed to support each test sample and position two thermocouples to measure rear surface temperatures. The design of the back plate enables the mounting of the sample to the IFH Sting arm that positions samples in the plasma stream.

Two test conditions were created in the IHF arc jet tests. Condition 1 was set to a heat flux of 48 W/cm^2 and condition 2 was set to 92 W/cm^2. The test protocol for the samples is given in Table 3 [22]. There are three arms called stings in the IHF: East, west, and overhead. Each arm/sample assembly can be sequentially rotated to position the center line of the sample in the center of the plasma jet.

Table 4
Arc Jet Test Samples*

Simulant	Formulation
JSC-1AC Lunar	Sintered "A"
JSC-1AC Lunar	Sintered "B"
JSC-1AC Lunar	Sintered "C"
JSC-1A Lunar	RTV
JSC-1A Lunar	RTV
JSC-1A Lunar	RTV
JSC-1A Lunar	RTV
JSC-1 Mars	RTV
JSC-1 Mars	RTV
JSC-1 Mars	RTV

Table 5
Arc Jet Test Protocol [20] [22]

Run #	Condition ID	Sting	Test Model	Model ID	Exposure (sec)	Model Instruments
1	1	OH	4" Slug Calorimeter	-	*	1 K TC
	1	E	JSC-1A Lunar/RTV	-	60	2 K TC
	1	W	JSC-1 Mars/RTV	-	90	2 K TC
2	1	OH	JSC-1 Mars/RTV	-	150	2 K TC
	1	E	JSC-1A Lunar/RTV	-	240	2 K TC
	1	W	JSC-1AC Lunar Sintered "C"	-	300	2 K TC
3	2	OH	4" Slug Calorimeter	-	*	1 K TC
	2	E	JSC-1A Lunar/RTV	-	60	2 K TC
	2	W	JSC-1 Mars/RTV	-	90	2 K TC
4	2	OH	JSC-1AC Lunar Sintered "A"	-	150	2 K TC
	2	E	JSC-1A Lunar/RTV	-	240	2 K TC
	2	W	JSC-1AC Lunar Sintered "B"	-	300	2 K TC

* Calibration exposure durations are per the discretion of the IHF test engineer.

Four test runs were performed over a period of two days. The calorimeter mentioned in Table 5 was used to set the heat flux (W/cm^2) and other system parameters for each test condition. The calorimeter consists of a copper iso-q model instrumented with thermocouples (TC) and was placed on one of the stings. Readings from the calorimeter allowed IHF personnel to tune the Arc Jet energy to the correct range for each condition. Test condition 2 is equivalent to the heating experienced by a space shuttle during re-entry of the Earth's atmosphere.

Front surface temperature of the samples was measured by three pyrometers. The pyrometers were requested to cover a surface temperature measurement range of 600-2500°C. The pyrometers were directed at the center of the each sample during and after the sample's exposure to the Arc Jet flow.

An infrared (IR) camera was focused to view the entire exposed face of the test article. All pyrometers and the IR camera were calibrated with a blackbody source. Also, a video camera was used to record each test. Pre and post-test still photos of the samples were taken. To

determine mass loss, each sample assembly (sample, back plate, thermocouples) was weighed both before and after the test. Photos of several samples under test in the arc jet are shown in Figs. 19-21.

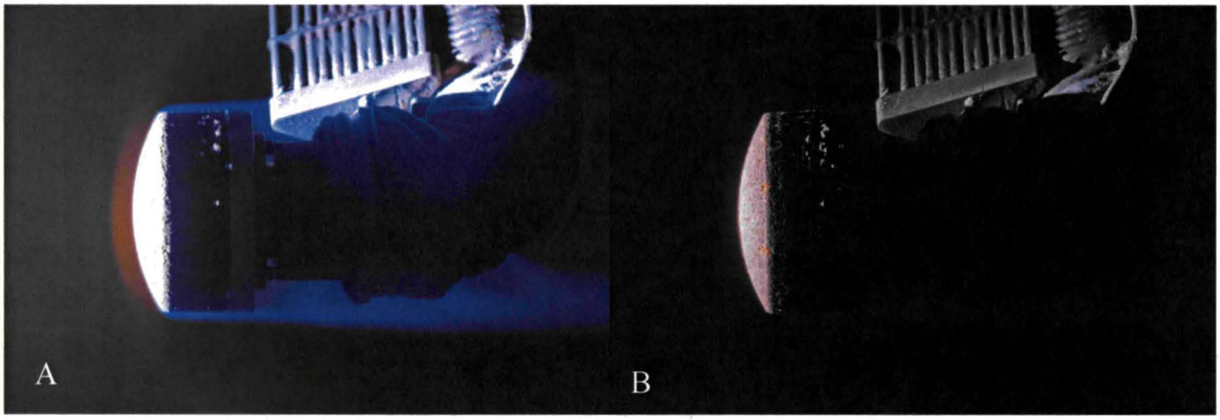


Figure 19. A. JSC-1 Mars/RTV at the start of run. B. The same sample at the end of a 150 second exposure at 48 W/cm^2 showing little or no melting or ablation.

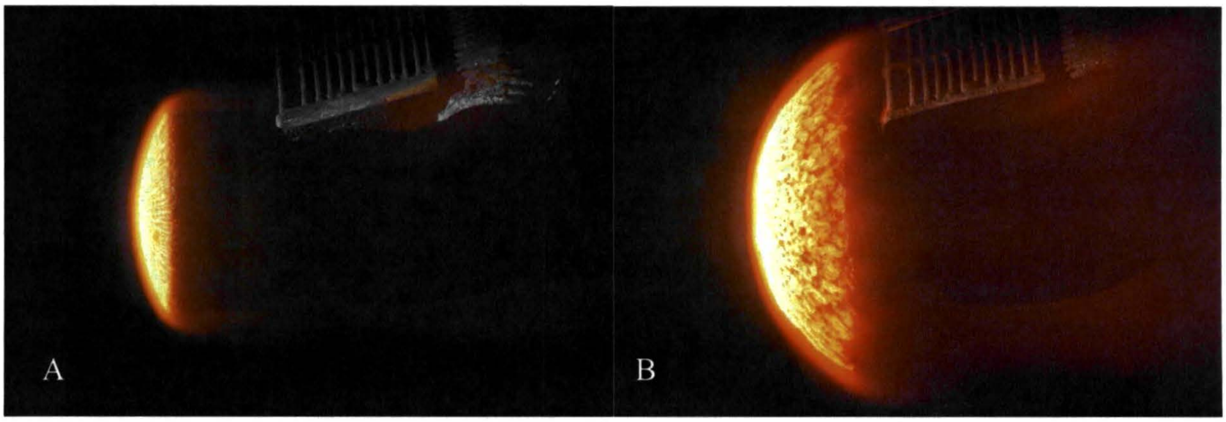


Figure 20. A. JSC-1A Lunar/RTV at the start of run. B. The same sample at the end of a 240 second exposure at 92 W/cm^2 showing melting and flow of its surface.

The highest recorded rear temperature was for sample JSC-1AC Lunar sintered "B" which was tested at condition 2 for 300 seconds (Fig. 21). While melting deformed and ablated most of the sample, it displayed sufficient thermal protection on its rear surface.

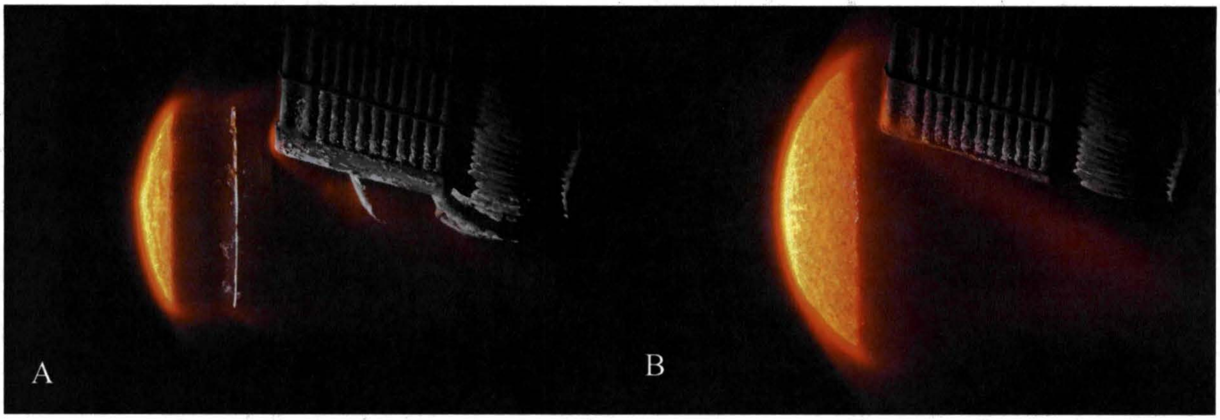


Figure 21. A. JSC-1AC Lunar Sintered "B" at the start of run. B. The same sample at the end of a 300 second exposure at 92 W/cm^2 showing melting and flow of its surface.

Mass loss for each of the sample assemblies (regolith sample, Aluminum back plate, thermocouples, bonding RTV) was acceptable. Except for the longer exposure times at condition 2 (92 W/cm^2) we measured little appreciable mass loss. Sample JSC-1AC sintered "B" displayed the most observable ablation and change. The melted regolith flowed over the surface of the sample and formed an approximately one inch wide and 3/8 inch thick glassy crust along the edges which was very brittle and broke off after the test. Pre-test and post-test photos of this sample are shown in Fig. 22. Also shown in Fig. 22 is a side photo of sample JSC-1AC sintered "A" showing a similar glassy ring. This ring, the result of surface melting and flow of molten regolith to the edges of the sample also changed the overall shape to a larger radius and enlarged the forward diameter by about two inches. Sample JSC-1AC Lunar sintered "B" also had the highest peak rear temperature recorded at 1237.7 seconds (987.7 seconds after test end).

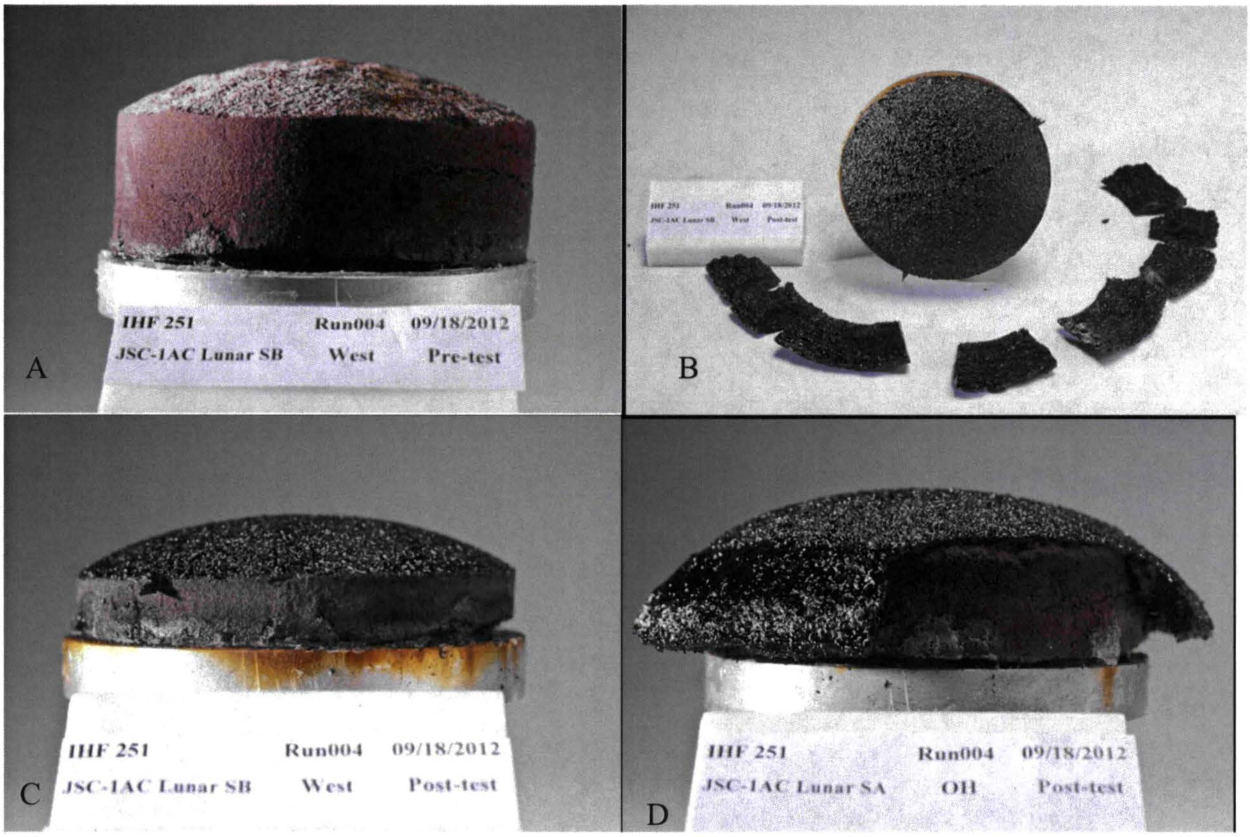


Figure 22. A. Pre-test side view of sample JSC-1AC sintered "B". B. Post-test photo showing glassy ring. C. Post-test side view showing loss of thickness in Sintered "B". D. Side photo of sample JSC-1AC sintered "A" showing an example of the glassy ring and its cross-section.

VII. Summary and Conclusions

Within the scope of the testing to date, the feasibility of using extraterrestrial regoliths as the construction material for atmospheric entry heat shields has been confirmed from the results of the acetylene flame and arc jet testing. While some of the arc jet-tested samples were heavily ablated, they provided adequate low temperatures on their rear surfaces. These rear surface peak temperatures were recorded several minutes after arc jet test termination.

While the highest energy input (92 W/cm^2) at a five minute duration was comparable to a space shuttle re-entry from low Earth orbit, interplanetary atmospheric entry energies can be on the order of about 300 W/cm^2 or higher [19][23][24][25]. For these type of atmospheric entries, a much thicker regolith derived heat shield would be required than the two inch thick samples evaluated. If the heat shield is fabricated on Phobos or an asteroid, where there is little gravity, then fairly large heat shields can be used to protect returning payloads to Earth.

A number of sintered and RTV bound formulations were evaluated. However, samples of JSC-1 Mars with the larger RTV concentrations performed well. In future work, JSC-1 Mars will be

replaced by a more representative simulant material using materials that represent characteristics of Phobos and Deimos materials instead. Other lunar material simulants that reflect anorthosite-rich highlands materials are also planned.

Much mission architecture work still needs to be performed to determine the full cost/benefit of using regoliths as atmospheric heat shield material. Cost savings for a 20-mission Mars campaign (10 unmanned and 10 manned missions) are estimated to be about $35 billion dollars if the massive heat shields for each mission did not have to be transported from the surface of the Earth to Mars [26].

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